FLIGHT PATH DESIGN ISSUES FOR THE TOPEX MISSION

J. C. Frautnick, E. Cutting

California Institute of Technology
Jet Propulsion Laboratory
Pasadena, California

Abstract

The proposed Ocean Topography Experiment (TOPEX) is an earth satellite mission currently under consideration by NASA. The primary purpose of the experiment is to determine the general circulation of the oceans and its variability. High precision, space based altimeter measurements will be combined with surface measurements and ocean models to accomplish the mission objectives. The paper will discuss mission requirements on orbit design, orbit selection space, derived requirements on navigation and satellite design issues which impact orbit selection. Unique aspects of the TOPEX orbit design are highlighted, such as high precision repeating orbits, "frozen orbit" values of eccentricity and periapses, precise maneuver and orbit determination requirements and insuring crossing arcs over a calibration site.

Introduction

The Ocean Topography Experiment (TOPEX) is a proposed NASA program to study the general circulation of the oceans by utilizing precision altimeter measurements from an earth orbiting satellite. This ambitious mission will require a carefully designed satellite system, precise orbit selection and maintenance, and a comprehensive mission design including ground truth and verification activities. This paper will discuss the flight path design issues for the TOPEX mission.

This paper is organized so that the basic objectives and requirements which drive the flight path design are presented first. Then the orbit selection space is defined and parametric analysis cf altitude, maneuver, eccentricity and inclination affects are summarized. A nominal orbit design is selected. The impact of orbit design on satellite design is then briefly discussed.

Science Requirements

The primary science objective of the Ocean Topography Experiment Program is to measure the surface topography of the ocean over entire ocean basins for several years, to integrate these measurements with subsurface measurements and models of the ocean's density field in order to determine the general circulation of the ocean and its variability. This information will then be used to understand the nature of dynamics, to calculate the heat transported by the oceans, the interaction of currents with waves and sea ice, and to test the ability to predict circulation from the forcing by winds.

The TOPEX satellite mission is the part of the TOPEX Program involving the satellite altimeter measurements. Figure 1 shows a possible configuration for the TOPEX satellite.

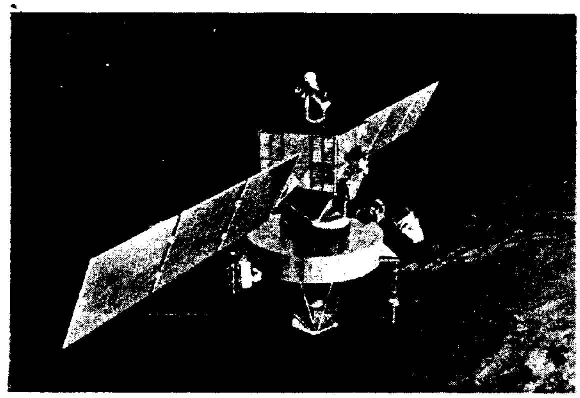

Figure 1. TOPEX Satellite

The TOPEX mission objectives are:

(1) Provide ocean topography measurements to enable calculation of the distribution of the mean and variable surface geostrophic currents.

(2) Process, verify and distribute altimetric and other geophysical data to Principal Investigators in a timely manner.

The scientific requirements necessary to meet the science objectives are given in Table 1.

The process by which satellite measurements can be used to measure surface currents is illustrated in Figure 2. The ellipsoid is a mathematical surface that approximates the shape of the Earth. The geoid is the surface the ocean would assume if there were no waves, tides or currents. The actual sea surface includes these dynamic effects. The orbit of the satellite is calculated very precisely by a "precision orbit determination" team. This process involves extensive calibration and in situ measurement to obtain near 10 cm accuracy. Once a precision orbit has been determined the satellite height above the reference surface can be calculated. Then adjusting for geoidal separation, mean sea level, earth and ocean tides and correcting for atmospheric affects on the altimeter signal and sea surface, and instrument delays will yield a height which can be compared against the raw altimeter height measurement. In areas, where the geoid is well determined, one can then determine the height of the ocean due to dynamic effects. If the geoid is not well determined, one can still determine the variability on repeating tracks.

Ocean currents cause a rise in the ocean surface above the geoid, such that the horizontal pressure gradient balances the Coriolis effect.

Table 1

Science Requirements

*1 Measure the topography of the sea surface with a precision of ±2 cm and an accuracy (1σ) of ± 10 cm (goal), ± 14 cm (requirement), with no geographically correlated errors, along a fixed measurement grid.

2 Measurements of ocean topography shall be made at least every 20 km along the subsatellite track.

3 Provide a means to sample the spatial and temporal variability of ocean surface currents with no aliasing of the geostrophic currents into semi-annual, annual or zero frequencies, or frequencies close to these frequencies.

4 Provide topographic measurements for a minimum of three years, with a capability to extend an additional two years.

5 Provide coverage of the oceans to at least as far south as the southern limit of the Drake Passage (62°).

6 Provide a means of repeating the measurements along a fixed grid every 10 days (nominal) and be capable of varying the repeat times between three and twenty days.

7 Provide a minimum of 81% of topographic data over the oceans and make data available to the principal investigators.
(Assumes the altimeter is always on over oceans)

8 Interim geophysical data records shall be made available prior to the final geophysical data records, to allow time to plan oceanographic experiments providing in situ observations simultaneously with satellite observations. Interim records shall use operational orbits (instead of precision orbits).

9 Provide a means to recover, correct, verify and process data to final form and deliver them to the Principal Investigators within 6 months of data acquisition.

* Requirement 1 applies for significant wave height less than 2m, wave skewness less than 0.1 and rainfall rate less than 2.0mm/hr.

The geostrophic flow equation [1] is:

$$u = \frac{g}{f} \tan \gamma \quad (1)$$

where u is the surface velocity of the current, g is the acceleration of gravity, f is the Coriolis parameter ($2\,\omega_e \sin\phi$ where ω_e is the rotation rate of the Earth and ϕ is latitude) and $\tan\gamma$ is the measured ocean surface slope normal to the current direction. For typical ocean currents the slope is very small (e.g. if u = 2 m/s at $\phi = 40°$, the slope is about 1 m in 50 km). Thus, the altimeter and the precision orbit determination must be very accurate. However, SEASAT results [2] have conclusively demonstrated that the technique is practical. It should be noted that only surface currents are determined by this method, and other

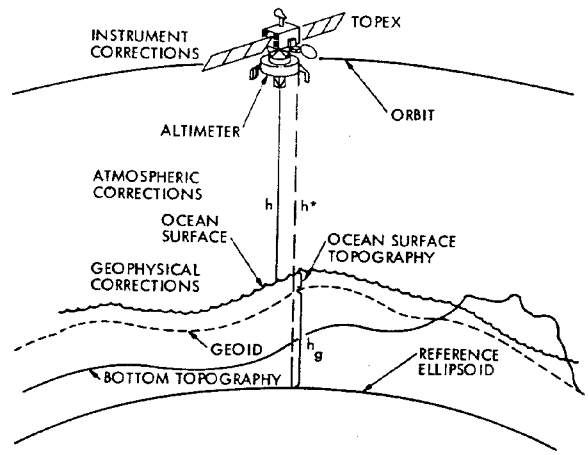

Figure 2. Sea Surface Height Measurements

observations (e.g., bathythermographs from ships) are required to determine the total flow.

Orbit Requirements

The orbit design requirements to meet the science requirements are presented in Table 2. Exact repeating orbits will allow changes in currents to be deduced from changes in altimeter measurements. The 1 km specification is an estimate of the distance over which a constant geoid can be assumed. The requirement on the orbit repeat period involves trading off short term variability (sampled by short repeat cycles) with the density of the ground tracks between repeat cycles (the longer the repeat cycle the more tracks are laid down). The 10 day nominal value is a compromise between these competing needs. Requirement 5 reflects the fact that uncertainties in the orbit knowledge must be kept very small to meet the science accuracy requirements. The requirement for ascending and descending ground traces over the calibration will allow for additional altimeter calibration data and also provide a check on orbit accuracy by differencing the range measurements. The other requirements are self explanatory.

Orbit Selection Space

The orbit design space for the TOPEX mission options study is shown in Figure 3. The figure shows inclination versus orbital precession rate with constant altitude contours shown. Areas labelled tides are altitude/inclination combinations which are unacceptable due to orbital resonance with oceanographic tidal effects.

There are two areas labelled 1 and 2 which include acceptable combinations of orbit elements. The boundaries of these areas are derived in the following pragraphs on altitude, maneuvers and inclination. A final paragraph discusses the relative advantages of area 1 and 2.

Altitude

Altitudes between 500 and 2000 km were examined. The upper limit on altimeter

Table 2

Orbit Design Requirements

1. The observational orbit shall be established and maintained such that the ground track crossing at the equator is with ±1 km of the desired longitude every repeat cycle
2. The orbit ground trace shall repeat exactly (±1 km) in a specified period between 3 and 20 days (nominally 10 days)
3. The orbit shall not recur at a period close to any tidal period or subharmonics.
4. Satellite subtracks shall cross at the equator with acute angles of 35 to 45 degrees.
5. The orbit altitude shall be high enough to reduce the influence of atmospheric drag and higher-order harmonics of the Earth's gravity field.
6. As a goal, the orbit shall have ascending and descending traces which intersect over the calibration site.
7. The orbit coverage shall include 62° latitude.
8. The orbit design shall be compatible with launch by either a Shuttle or possibly by a Delta launch vehicle from WSMC.

performance is assumed to be 1500 km. This limit is derived from the required altimeter transmitter power, practical limits to antenna size and footprint and beam width considerations.

The lower limit is derived from the minimum maneuver frequency interval. The maneuver frequency analysis is shown in the next section. Altitudes at 1334, 1000 and 800 km were selected as study options.

Exact repeat orbit altitudes are shown in Figure 4. For TOPEX, a repeat of approximately 10 days is required(Science Requirement 6). The number in the figure indicates the number of revolutions the satellite makes before exactly repeating its ground trace. The figure is derived using 2-body motion with J_{20} effects only. The second axis on the abscissa is S, the fundamental interval. S is the distance between consecutive ascending nodes in an earth-fixed frame.

$$S = P_n (\omega_e - \dot{\Omega}) \quad (2)$$

where P_n = nodal period

$\dot{\Omega}$ = inertial nodal precession rate

ω_e = earth rotation rate

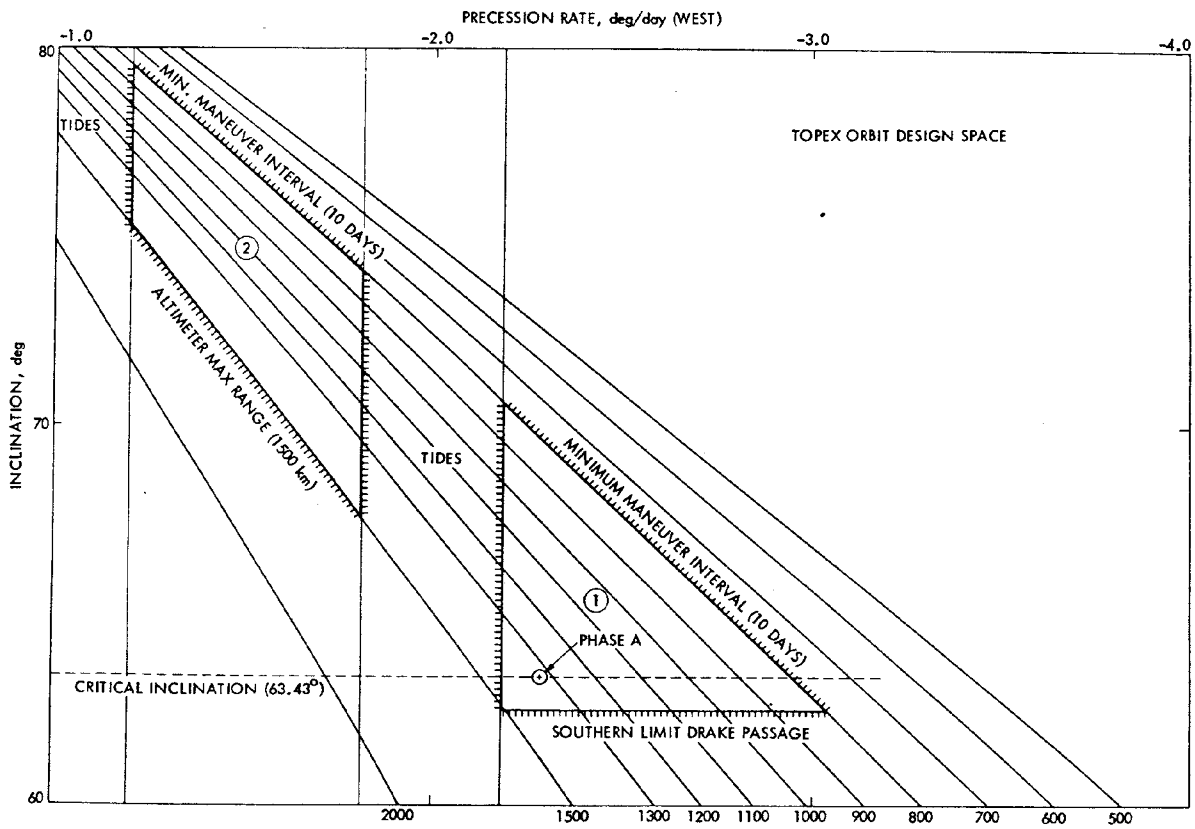

Figure 3. TOPEX Orbit Design Space

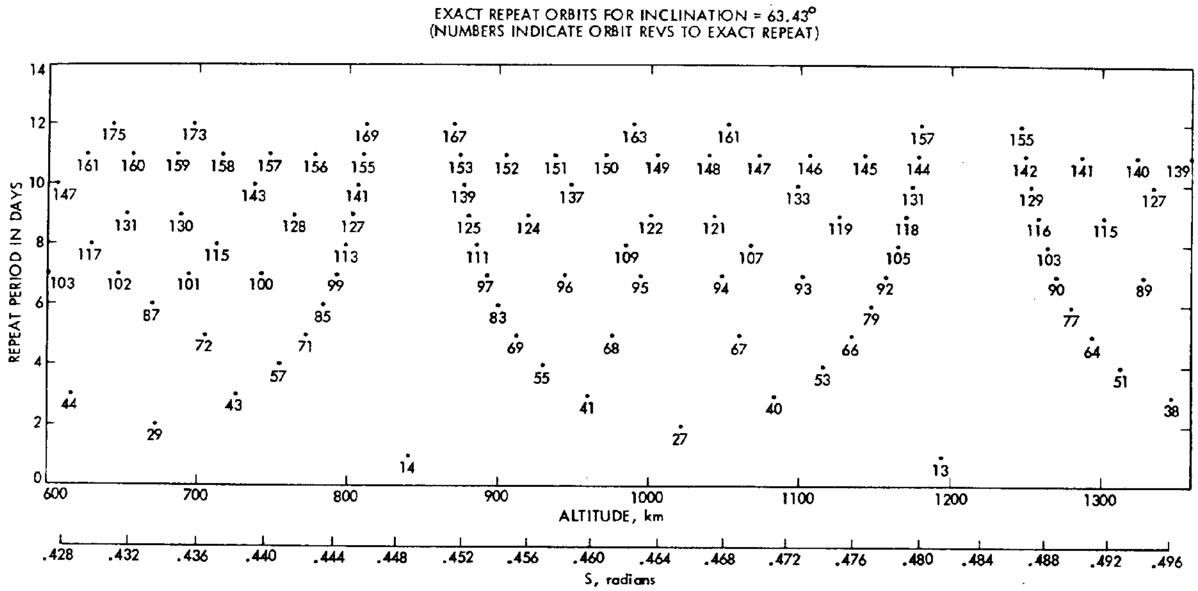

Figure 4. Exact Repeat Orbit Altitudes

For the comparison, the following 10 days exact repeat orbits were selected:

Altitude	Revolutions
1334	127
948	137
806	141

Note in Figure 4 that many 11 day and 9 day repeat orbits exist at intermediate altitudes.

Maneuvers

Maneuvers are required to maintain the 10 day repeat orbit ground traces to within ± 1 km and keep eccentricity $\leq .001$. Maneuvers may also be made to change a repeat cycle or to adjust ground trace phasing.

It will be shown in the following paragraphs that the orbit altitude impacts primarily maneuver frequency while the eccentricity control requirement impacts the total ΔV required.

Semi-Major Axis Effects

The mean semi-major axis, a, is perturbed by atmospheric drag, gravitation effects, solar radiation pressure and other smaller forces. Drag decreases the semi-major axis and causes errors to build up in the exact repeat pattern. Eventually the ± 1 km constraint will be exceeded and a corrective maneuver will be made to the correct semi-major axis. The amount of drag is dependent on the atmospheric density and the satellite area/mass ratio. Atmospheric density is a function of altitude and is affected by solar UV and EUV radiation, which heats the atmosphere.

Figure 5 illustrates the long term 11 year solar cycle. The smooth curve is a moving average and the connected points are monthly averages. Note that the solar activity rises relativity fast and falls slow. Daily values generally range from 1/2 to 1-1/2 the monthly average. Assuming an 11 year cycle, the next solar minimum is 1987 (± 1 year). By adding 11 years to solar cycle 21 (beginning in 1976) we can grossly estimate conditions for TOPEX. Assuming a January '89 launch would predict initially low solar activity. However, 2-3 years later, solar flux could rise to an average of 200 with a monthly average over 220. If we assume the next solar cycle is closer to cycle 20 than 21, the maximum monthly average is closer to 160. For purposes of estimating maneuver frequency, flux values of 100, 150 and 200 have been assumed. The lower value is expected near launch while the other values may occur in subsequent years.

Table 3 is an estimate of maneuver frequency as a function of altitude, satellite area/mass, and 10.7 cm solar flux values. The number of maneuvers was estimated by assuming 1 mm/sec execution error and 5m uncertainty in operational orbit determination. (Precision OD will of course be much more accurate but its time delay makes it unavailable for operational decisions). The estimated maneuver frequency is determined by the technique developed by Kechichian [3]. (Initial

Table 3

"Estimated" Maneuver Frequency

Average number of days between maneuvers

ALTITUDE (KM)	SOLAR FLUX 100	150	200	AREA/MASS (M^2/KG)
700	14	2.8	0.8	.010
800	40	9	3.8	
1000	57	50	14	
1334	547	273	88	
700	7	1.4	0.4	.020
800	17	4.5	1.2	
1000	34	22	6	
1334	304	103	39	

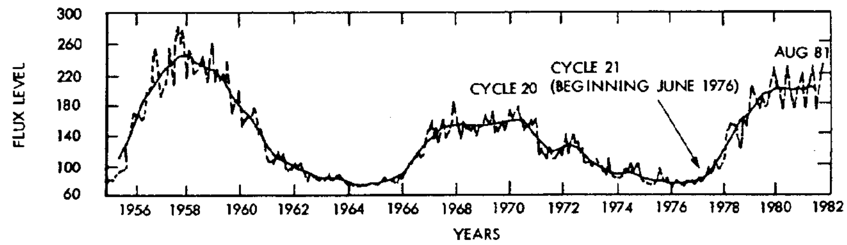

Figure 5. Long-term Solar Cycle Variation

westward node drift to initially offset drag effects.)

The highest acceptable maneuver frequency from a maneuver operations standpoint is 1 to 2 weeks, on a regular basis. The Precision OD Team has independently recommended that maneuver frequency be larger than the repeat cycle (currently 10 days) to allow uninterrupted data gathering. Consequently, a value of 10 days has been adopted as a minimum maneuver frequency. By this criteria, orbits at 700 km altitude are unacceptable and orbits of 800 km are marginal. 800 km altitude has been adopted as the minimum acceptable altitude although maneuver frequency may be more often than 10 days near 1990.

It should be noted that even when the mean time between maneuvers is large, there is some probability that individual maneuvers may occur close together in time. This is due to combinations of OD error and execution error (e.g. satellite actually higher than thought and maneuver execution larger than needed). Such combinations are somewhat unlikely but are significantly probable to plan for. Figure 6 shows a rough estimate of the distribution of time between maneuvers with a mean value of 59 days (1000 samples). Note in Figure 6 that many samples were <30 days and there were some <10 days.

However the most important consideration in evaluating altitude choices turned out to be the accuracy available from the Precision Orbit Determination. As noted earlier, the first science requirement calls for 14 cm (10 cm goal) accuracy in measuring topography. Covariance analyses were done for 800, 1000 and 1334 km altitude orbits utilizing one-way doppler tracking from 45 upgraded TRANET-II tracking stations [4]. The orbit error budgets were 26, 18 and 13 cm respectively for the three altitudes. The errors were primarily due to uncertainty in the gravity field. This result, more than any other, has forced the TOPEX altitude to 1334 km.

Eccentricity Effects

It is desired to minimize drift in eccentricity in order to minimize altitude variation corrections for the altimeter. Mean eccentricity (e) will be kept $\lesssim .001$. Eccentricity is affected primarily by the earth's zonal harmonics:

$$\frac{de}{dt} = J_2 0(e) - \frac{\frac{3}{2} nR^3 J_3 \sin i}{a^3(1-e^2)^2} \left(1-\frac{5}{4}\sin^2 i\right)\cos\omega + J_4 0(e) + \ldots.. \quad (3)$$

Note that for i = 63.43 and e = 0, all terms are 0. However, higher order odd zonal terms introduce a secular drift to e of about $7x10^{-6}$ /day (at 1334 km) see Figure 7. This rate requires a compensating maneuver each 142 days, or 7 maneuvers in 3 years. In actual practice the eccentricity maneuvers would be combined with the

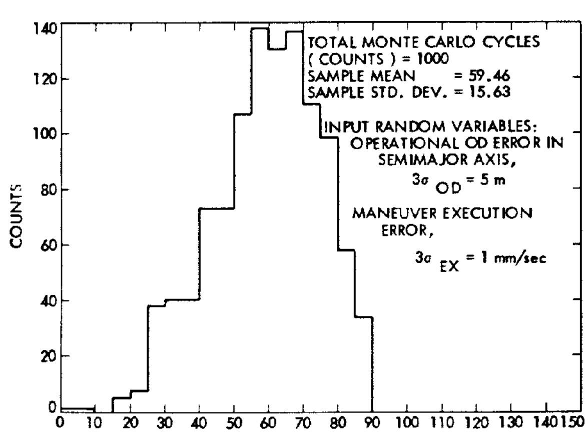

Figure 6. Histogram of Time Between Maneuvers

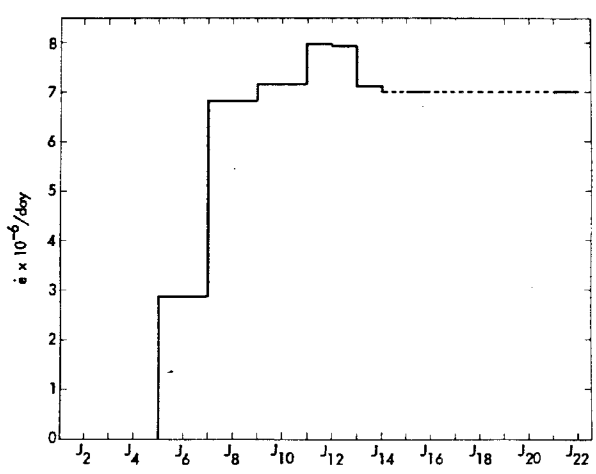

Figure 7. Effect of Zonals on Eccentricity Rate

semi-major axis corrections. Thus the decay of semi-major axis determines the maneuver frequency. However, since the eccentricity correction is larger, eccentricity rate determines the total ΔV required. Semi-major axis corrections can be done along with eccentricity corrections for no additional fuel by choosing the appropriate maneuver locations. This was the technique used for Seasat [5].

Lower altitudes have the effect of causing a slightly greater rate of $\dot{e}$ and thus a larger ΔV is required. These findings are shown in Table 4. Since the ΔV requirements are small in all cases, there is no significant impact of altitude on maneuver ΔV.

Table 4

Five Year ΔV Requirements
(orbit maintenance only)

ALTITUDE(KM)	ΔV
800	77 m/s
1000	66 m/s
1334	46 m/s

Inclination Effects

Inclination is a key parameter in TOPEX orbit design. The requirement (see Table 1) for ocean coverage at least as far south as the southern limit of the Drake passage, 62° south, yields the following limits on inclination

$$62^\circ \lesssim i \lesssim 118^\circ$$

Nodal precession rate, $\dot{\Omega}$, depends on inclination and is important in the tidal aliasing constraint discussed below.

$$\dot{\Omega} = -\frac{3}{2}\, n \left(\frac{R_\oplus}{p}\right)^2 J_{20}^2 \cos i \qquad (4)$$

$$n = \left(\frac{\mu_\oplus}{a^3}\right)^{1/2} \qquad (5)$$

$$p = a(1-e^2) \qquad (6)$$

n is the mean angular motion and p is the "parameter" of the orbit. $\mu_\oplus$ is the gravitational constant of the Earth and J_{20} is the second zonal harmonic of the Earth. $\dot{\Omega}$ is positive to the east (retrograde orbit).

The TOPEX tidal aliasing constraint (Table 1) requires that the satellite ocean track not repeat in phase with a major tidal component. Suppose that the track did repeat at some close submultiple of a tidal frequency. Then, successive altimeter observations over the same point would indicate a slowly varying ocean height. The tide effect could be confused with an ocean current. This frequency ambiguity effect in sampling systems is called aliasing. The apparent shift in frequency is called the aliasing frequency ω_a. For TOPEX

$$\omega_a = |\omega_t - N(\omega_e - \dot{\Omega})| \qquad (7)$$

where ω_t = the frequency (deg/day) of some tidal component and N=1 for a durnal tide and N=2 for a semi-diurnal tide. Values of ω_t are available [6].

Figure 8 (from R. Stewart,[7]) shows a plot of ω_a vs $\dot{\Omega}$ for various tidal components. Values of $\omega_a \approx$ zero are to be avoided since these tides will look like steady or slowly varying currents. Values of $\omega_a \approx 1$ are also troublesome, because many currents have a strong annual variability. It is seen from Figure 8 that prograde orbits ($\dot{\Omega}$ negative, i<90°) are the best choice. Values near $\dot{\Omega} = -1$ must be avoided because K_1 aliases into 1 cycle/year. How close one may come to $\dot{\Omega} = -1$ depends on the resolution of the Fourier spectrum of the data, which in turn depends on data record length, data noise, etc. In order to resolve tidal components, the region near $\dot{\Omega} = -2$ must be avoided since P_1 and K_2 alias into the same frequency. Figure 3 shows the excluded bands around $\dot{\Omega} = -1$ and -2.

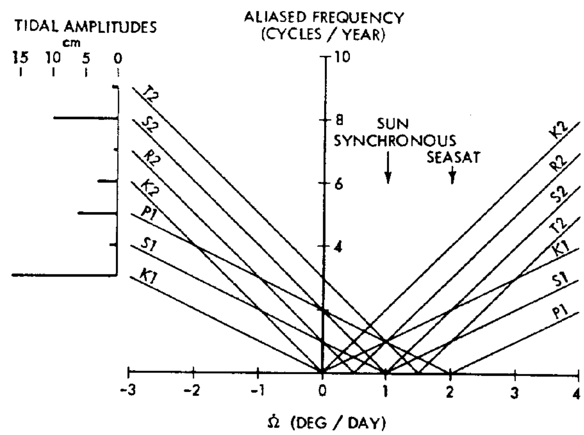

Figure 8. Aliasing of Tidal Components

For purposes of altimeter calibration, it is desired that an ascending and a descending ocean trace intersect over the Bermuda laser tracking site (Table 1). Figure 9 shows double crossings as a function of altitude and inclination. Each point on the curves is also a 10 day exact repeat orbit, so the double crossings recur every 10 days. Curves labeled "☊1st" are ascending crossing first; "☋1st" means descending first. N is the number of revolutions between ascending and descending crossings. (It is desired to minimize the time lapse between intersecting passes so observation conditions are similar). At 1334 km, double crossings are available at 62.5 and 68.0 inclination. A technique for determining crossing latitude is given by J. King [8]. Figure 10 shows an example of a double Bermuda crossing. Crossing longitude can be adjusted as needed by phasing maneuvers.

It is required that ocean trace crossing angles be in the range 35°-45° at the equator (Table 2). The ocean surface current, u, normal to the satellite track, is determined from Equation 1.

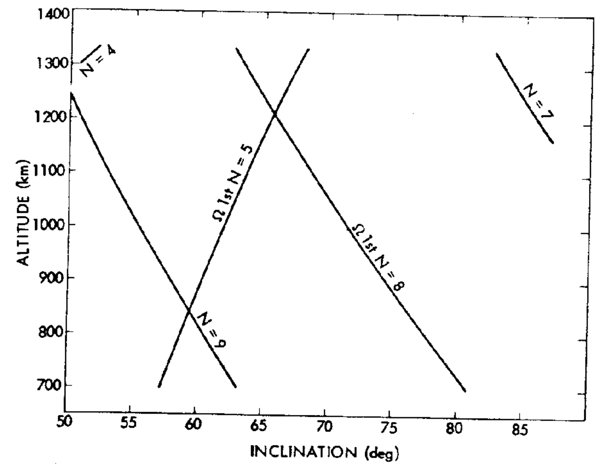

Figure 9. Bermuda Crossings, Altitude vs. Inclination (N = Revs between crossings)

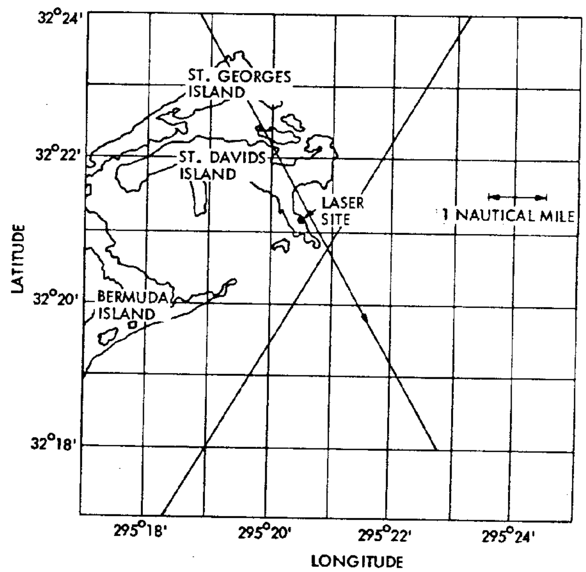

Figure 10. Example of Bermuda Overflight

Crossing angles in the range 35°-45° give a good determination of two components of surface current. Further, there is a good intersection with western boundary currents at mid-latitudes. The crossing angle, Ψ, at the equator is

$$\psi = 2\ \mathrm{Tan}^{-1}\left|\frac{V_s \cos i - V_e}{V_s \sin i}\right| \qquad (8)$$

where $V_e = \omega_e R_\oplus$ and V_s is the speed of the subsatellite point on the surface of the Earth, if the Earth were nonrotating. At 1334 km altitude, the equator crossing angles vary from 52° at inclination of 60° to 27° at inclination of 72°. (For SEASAT, a retrograde orbit was used to maximize ψ, [5].

Inclination also affects the rate of drift of orbital parameters due to the second and third zonal harmonics of the Earth, J_{20} and J_{30}. The rate of change of eccentricity, $\dot{e}$, was given earlier in Equation (3). The rate of change of argument of perigee, for small values of e is

$$\frac{d\omega}{dt} = \frac{3nR_\oplus^2}{a^2(1-e)^2}\ J_2\left(\frac{5}{4}\sin^2 i\right) \qquad (9)$$

$$-\ x\left[1+\frac{J_3 R_\oplus}{2J_2 a(1-e^2)}\ \ \frac{\sin^2 i - e\cos^2 i}{\sin i}\ \ \frac{\sin\omega}{e}\right]$$

A nominal inclination value of 63.43° was initially selected because this value makes the mean Eccentricity rate of change, $\dot{e}$, equal to zero to first order (see Equation 3).

However, as shown earlier, Equation 3 can be expanded to include higher order harmonic terms. These higher order terms, specifically the odd zonal harmonics through J_{13}, cause a secular growth in e, see Figure 11. Note in Figure 11 that the eventual growth in e is independent of the initial value of perigee. The secular growth in e is the driving factor in the ΔV budget.

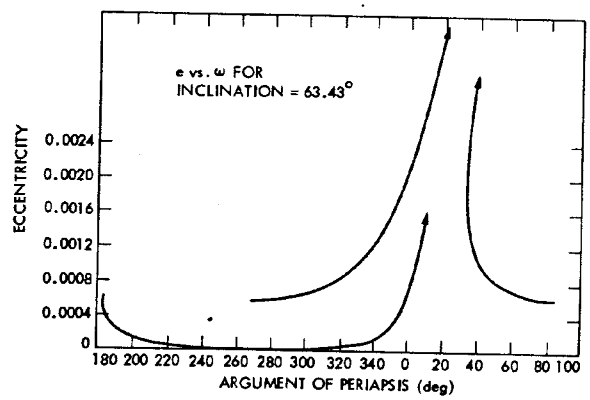

Figure 11. e vs. ω for Inclination = 63.43 Degrees

Higher values of inclination can produce some interesting variations in e and ω behavior. Figure 12 shows how e and ω behave for inclinations between 64.5° and 70°. The inclinations between 64.5 and 65.3 allow e and ω to librate naturally about a null point (value of e and ω that produce a zero time rate of change). The null point value of ω is 270° and the null point value of e decreases as inclination increases. Near i = 66°, the null point value of ω shifts to 90°. Above i = 66°, the null point value of e increases with i. It is concluded that for small values of e (≤ .001), and i > 64.7°, values of e and ω can be chosen such that neither will experience a secular increase.

The idea of selecting e and ω to minimize e and ω variation was utilized on Seasat. It has the advantage of minimizing fuel since changes in e drive the orbit maintenance ΔV budget. However

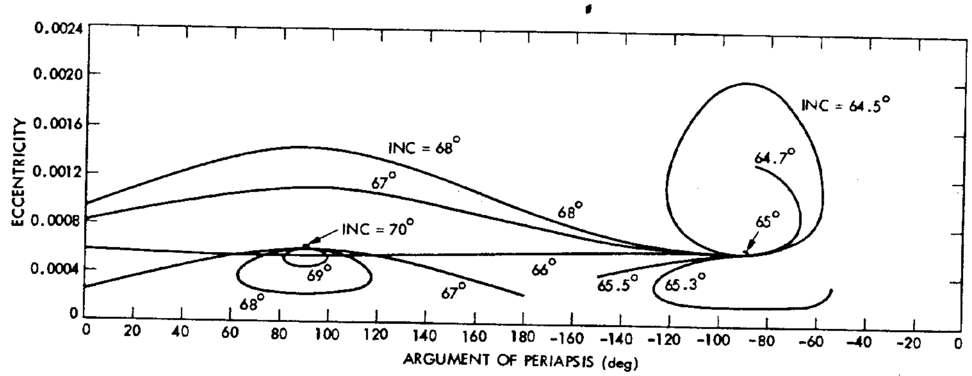

Figure 12. e vs. ω Behavior

the savings in ΔV is only a few 10s of meters/second, and has no associated cost savings. There is slight advantage in simpler maneuver operations if e and ω are basically self controlling.

In summary of the inclination studies, inclinations near 63.43° balance the disturbing perturbations of the earth's J_{20} and J_{30} zonal harmonics. However, eccentricity is subject to a secular growth due to higher order zonal harmonics. Previous work done on Seasat has shown that for small, but non-zero, values of e,e and the mean argument of perigee will librate about a stable point. This effect is most pronounced at inclinations away from 63.43° since J_{20} and J_{30} must balance each other. Note that the farther from 63.43°, the lower the range of e and the smaller the value that will set e = 0. However, since fuel to correct e is not a major issue, the main benefit of this orbit tuning would be to minimize maneuvers and simplify mission operations. A small cost savings would occur.

Ascent

The Shuttle will deliver TOPEX into a circular orbit at 278 km (150 n.mi) altitude at the desired inclination. The orbital transfer from the Shuttle orbit to the TOPEX observational orbit is called the ascent phase. (An alternate launch vehicle possibility for TOPEX is the Delta which could deliver TOPEX directly to the observational orbit.) The TOPEX ascent propulsion system provides the necessary impulse to transfer between coplanar circular orbits at 278 km and, say 1334 km. In order to maintain flexibility for satellite design, both high thrust and low thrust ascent propulsion systems are considered. The low thrust system is typically a monopropellant blowdown system with initial acceleration of .05 g's, decreasing as propellant is consumed.

For the high thrust option, the Hohman transfer impulse is 549 m/sec. Several thrust profiles are possible for the low thrust option: (1) continuous low thrust burn, (2) low thrust with a single coast period, (3) multiple low thrust burns at successive perigees and apogees. The multiple burn case approaches a series of incremental Hohman transfers and the total impulse approaches the Hohman impulse, as the number of burns increases. However, multiple burns introduce unreliability and operational complexity.

For operational simplicity, it is also desired to perform the ascent in an open loop guidance mode. That is, burns and steering programs are executed based on precomputed, stored commands. Of course, errors will exist in the final orbit due to thrust magnitude and pointing errors. Studies have shown that the ascent errors are small enough to be corrected by the orbit maintenance system.

Recommended Selection Space

The boundaries to the acceptable areas of the orbit design space have been discussed in the preceding paragraphs. Figure 3 shows two acceptable areas: 1 and 2. Area 1 is preferred due to the larger crossing angles at the lower inclinations. Also, double Bermuda crossings are not available for 82° > i > 72° if altitude is between 1000 and 1350 km. Therefore area 1 is recommended as the orbit design space. Area 2 is acceptable and should be considered if there are significant cost benefits to be gained in that region.

Baseline Orbit

The baseline orbit used during 1982 studies was the Phase A baseline orbit shown in Table 5. All parameters lie within the orbit design space in Figure 3. Note, however, that the inclination of 63.43° does not minimize e (See inclination Section) nor does it allow for double Bermuda overflights. It is anticipated that work in 1983 will re-evaluate the baseline orbit value of inclination.

Table 5 TOPEX Baseline Mean Orbital Elements

a	=	7712.1903 km	(semimajor axis)
e	=	0	(eccentricity)
i	=	63.434949°	(inclination)
Ω	=	277.67199°	(node right ascension)
ω	=	0°	(argument of perigee)
M	=	0°	(mean anomaly at epoch)
epoch	=	1 September 1986	13 hrs 46 min 08 sec GMT

The Baseline Orbit has the following key characteristics. It is circular at a mean equatorial altitude of 1334.05 km. The nodal period is 112.36 minutes and the orbital speed is 7.189 km/s. The longitude of the ascending node repeats exactly after 127 revolutions or about 10 days. The spacing between ascending nodes is 28.35° The orbit plane precesses at an inertial rate of 2.29°/day to the West. (The Baseline Orbit repeats exactly assuming J_{20} perturbations only.) The initial ascending node will be chosen to assure an initial full sun orientation of the satellite.

Interaction of Orbit Design and Satellite Design

The flight path design and the satellite design for TOPEX interact in some interesting ways. The satellite area to mass ratio, A/m, affects the choice of orbit altitude. The orbit altitude is desired to be high enough to minimize orbit determination errors due to uncertainties in drag, solar pressure and gravity harmonics, since orbit determination errors affect topographic accuracy. However, the altitude must be low enough that there is sufficient signal return for the altimeter. Note that A/m may increase with altitude as spacecraft design changes to supply the extra power (solar panel area) and antenna gain (dish diameter) needed by the altimeter. Tradeoff studies indicate satellite A/m in the range .010-.020 m^2/kg will provide adequate performance for altitudes in the range 1000-1500 km.

The design of the satellite power system depends on the solar geometry which is determined by the orbit design. The tidal aliasing constraint dictates an orbit which is not sun synchronous, so there will be periods of sun occultation. Solar panel area, orientation and battery size must be compatible with the occultations and with the expected variation in solar angle during the mission. Satellite thermal design is also affected by sun geometry.

The satellite will probably require a steerable, high gain antenna in order to communicate with the geostationary Tracking Data Relay Satellites (TDRS). Antenna gain and pointing angles are determined from TOPEX orbit geometry with respect to TDRS. The timing error of the satellite clock affects the orbit design. Orbit eccentricity, e, must be controlled so that e<.001, in order that the product of timing error and altitude rate have a negligible contribution to topographic error.

Ascent propulsion system velocity requirements and possible mechanizations have been discussed earlier. The on-orbit control requirement of ± 1 km, coupled with the desire to maximize the time between maneuvers, imposes some special requirements on the orbit maintenance propulsion system. The thrusters must be able to make small (e.g. 1/2 m/sec) maneuvers with great precision, .001 m/sec [3]. The fuel load for orbit maintenance depends on ascent errors, on orbit perturbations, on orbit prediction error and thruster execution errors, and on adaptive maneuvers needed to enhance science. As TOPEX satellite design advances, the interactions with orbit design will be studied in more detail.

Conclusions

It has been shown that the flight path can be selected from a range of values depicted in the orbit design space, Figure 3. The following conclusons are drawn concerning flight path orbit parameters:

- Orbit altitude should be between 800 and 1400 km.
- Orbits <800 km altitude require excessive maneuvers (unless a drag compensation system is considered).
- Maneuver frequency can be minimized by: going to higher altitudes, making the Bermuda miss distance as large as possible, reducing satellite A/m, and launching at or before solar minimum.
- Altitude decay rate drives maneuver frequency.
- Eccentricity rate drives ΔV required.
- Inclination can be selected to minimize e and provide double Bermuda Island crossings.

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